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Science Applications, Inc.
Rolling Meadows, Ill.

Advanced Planetary Studies

Second Annual Report

STAR Abstract

Results of planetary advanced studies and planning support provided by Science Applications, Inc. staff members to the Planetary Programs Division of OSS/NASA for the period 1 February 1974 through 31 January 1975 are summarized. The scope of analyses includes cost estimation research, planetary mission performance, Shuttle planning, Jupiter orbiter lifetime assessment, Titan mission concepts, penetrator deployment, and advanced planning activities. This work covers 4 man-years of research. Study reports and related publications are included in a bibliography section.

Report No. SAI 1-120-194-A2

ADVANCED PLANETARY STUDIES
SECOND ANNUAL REPORT

by

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for

Planetary Programs Division
Office of Space Science
NASA Headquarters
Washington, D. C.

Contract No. NASW-2613

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FOREWORD

This report summarizes the results of planetary advanced studies and planning support performed by Science Applications, Inc. (SAI) under Contract No. NASW-2613 for the Planetary Programs Division, Code SL, of NASA Headquarters during the twelve month period 1 February 1974 through 31 January 1975. A total effort of 7760 man-hours (47.9 man-months) was expended on six specific study tasks and one general support task. The total contract value was $207, 748, with 93% of the work performed by the staff of the SAI Chicago office. Inquiries regarding further information on the results reported here may be directed to the project leader, Mr. John Niehoff, at 312/253-5500.

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1. INTRODUCTION

Science Applications, Inc. (SAI) participates in a program of advanced concepts studies and planning analysis for the Planetary Programs Office, Code SL, of NASA Headquarters. SAI's charter is to provide unbiased preliminary analyses and evaluations for Code SL planning activities. Specifically, the objective of this support is to ensure that NASA has an adequate range of viable future planetary mission options in order to pursue the objectives of solar system exploration within the changing constraints of our space program. The nature of the work involved is quite varied, ranging from short quick response items to pre-Phase A level mission studies. During the past contract year a total of ten SAI staff members contributed to this effort.

The purpose of this report is to summarize the significant results generated under this advanced studies contract during the twelve month period, 1 February 1974 through 31 January 1975. Progress reports of the task efforts have been given at scheduled quarterly reviews, and in Code SL's Quarterly Newsletter. Task reports have been prepared and presentations given to a wide audience at NASA Headquarters, NASA Centers, and at technical meetings on the significant study results. This report, therefore, is necessarily brief, with the intention of directing previously uninformed interested readers to detailed documentation, and to serve as a future reference to previously completed advanced studies.

The next section of the report presents the individual tasks performed during the contract period and briefly describes each task presenting the key results and conclusions that were generated. The last section of the report is a bibliography of the reports and publications that have resulted from the task analyses. SAI is presently beginning another twelve month period of advanced studies for the Planetary Program Division with a schedule of eight study tasks, several of which are continuing research on the work reported here.

2. TASK SUMMARIES

An initial schedule of six study tasks was planned for the twelve month contract period, 1 February 1974 through 31 January 1975. Two additional tasks were added during the contract period bringing the full schedule to eight tasks. These tasks, listed by contract task number, are as follows:

1) Cost Estimation Research,
2) Planetary Missions Performance Handbook - Vol. I, Outer Planets,
3) Shuttle Impact Planning,
4) Jupiter Orbiter Lifetime Analysis,
5) Titan Mission Concepts Study,
6) Advanced Planning Activity,
7) Error/Control Analysis of Penetrator Deployment at the Moon and Mercury,
8) Expenditure Management of the Symposium on Outer Planet Exploration.

Task 6, Advanced Planning Activity, is a general support task designed to provide a budgeted level of effort for technical assistance on short term planning problems which frequently confront the Planetary Programs Division. The remaining first seven tasks are planned efforts with specific objectives of analysis. Task 7, Error/Control Analysis of Penetrator Deployment at the Moon and Mercury, was added in the eigth month of the contract period to replace continued effort on Task 3, Shuttle Impact Planning, which was prematurely terminated to await results of the Shuttle Interim Upper Stage (IUS) contractor definition studies. Task 8, Expenditure Management of the Symposium on Outer Planet Exploration was added in the tenth month to facilitate the formulation of this symposium on outer planet mission planning strategy. No

technical manpower was involved in performing this service. Hence, no further discussion regarding this task is provided.

A total of 7760 man-hours of effort (47.9 man-months) was expended in completing this schedule of tasks. A brief description and presentation of key results for each of the first seven tasks is presented in the subsections which follow. The level of effort devoted to each of the tasks is given with the task title at the beginning of each subsection. Specific reports generated as part of the study tasks are noted, with a complete list of publications given in Section 3 of the report.

2.1. Cost Estimation Research (1940 man-hours)

This task is continuing research in the development of a planetary mission cost estimating model. The purpose of the model is to provide reasonably accurate rapid estimates of future planetary missions for planning activities of the Planetary Programs Division. Historically, cost estimates of future missions have been in error (underestimated), in extreme cases, by more than 100% with 50% errors not uncommon. Also, the errors in the estimates were often related to the complexity of the mission, with soft landers being most poorly estimated. The need for an objective systematic approach for generating reasonably accurate initial estimates of advanced mission concepts which would be sufficiently reliable for scheduling future projects to budgeting guidelines was the genisis of the Cost Estimation Research Task.

The cost model being developed under this task has a stated accuracy goal of $\pm$ 25% error on the estimate. The model is applicable to a wide range of planetary mission types including flybys, orbiters, atmospheric entry probes and soft landers. The model input requirements have been restricted to pre-Phase A level definitions because the generated project estimates are for future mission concepts. A complete

list of all possible model input parameters is presented in Table 1. The present cost model structure is functionally summarized in Figure 1. Several of the characterizing features of this cost model are apparent in Figure 1. First, the basic cost unit of estimation used by the model is man-hours. For severely limited hardware production projects, such as planetary missions, manpower, in units of direct labor hours, is the key element of cost. The low volume production characteristic and the NASA cost reporting system were found to stabilize direct labor cost at 29.6% of total project cost for a wide range of mission concepts analyzed with a very small standard deviation of 1.3%. Estimating manpower, rather than dollars has the following benefits: 1) simplifying the actual estimation procedure, since fewer cost elements are involved; 2) removing the effects of inflation from the estimating procedure; and 3) providing added visibility to the cost reduction effects of learning and inheritance.

The second characteristic of the model is that basic estimation is done at the subsystem hardware level with subsequent estimates of support functions and non-labor costs being built upon these values. Although the cost data base upon which the labor estimating relationships (LERs) were and are being developed provide resolution down to component hardware levels, the LERs themselves begin at the subsystem level so that input requirements do not exceed the information content of pre-Phase A mission studies. The actual input, described in Table 1, is composed largely of subsystem masses and key mission event times for much the same reason.

The third key characteristic of the model is its ability to allow for the cost benefits of direct inheritance from recent projects utilizing similar or identical subsystems. At present the inheritance modelling is applicable to ad hoc opportunities of hardware and design inheritance but the procedure is sufficiently general to permit the inclusion of cost benefits from standardized hardware at some future date.

Table 1

COST MODEL INPUT PARAMETERS

Z$	Date of First Launch (Month & Calendar Yr. e.g., 11/1975)
D2	Fiscal Wage Date (Fiscal Yr. e.g., 1975.9)
N1	Number of Flight Articles
W1	Weight of Power Subsystem Excluding RTG's (lbs.)
N2	Number of RTG Units per Spacecraft
L1	RTG Fuel Loading (Thermal Watts)
S1	Total Weight of Structure Subsystem (lbs.)
S2	Weight of Mechanisms and Landing Gear (lbs.)
S3	Weight of Thermal Control, Pyro. and Cabling (lbs.)
P1	Propulsion System Dry Weight Excluding Throttable Liquid Vernier for Landers (lbs.)
P2	Liquid Vernier Dry Weight (lbs.)
P3	Aerodeceleration Subsystem Weight (lbs.)
G1	Total Weight of Guidance/Control Subsystem (lbs.)
G2	Weight of Radar in G/C Subsystem (lbs.)
C1	Weight of Radio Frequency Comm. Subsystem (lbs.)
C2	Weight of Data Handling Subsystem (lbs.)
C3	Weight of Antennas (lbs.)
Q1	Total Weight of Science Experiments (lbs.)
Q2	Weight of Lander Surface Experiments in Q1 Having Significant Sampling/Processing Operations (lbs.)
Q3	Pixels per Line of TV (or Equivalent Visual Imaging)
K1	Total Mission Duration From First Launch to End of Last Minus Time When No Spacecraft is in Flight (mo)
K2	Total Encounter Time of the Prime Mission (mo)
K3	Total Number of Encounter Phase Start Ups
K4	Total Number of Science Teams During Encounter Phase

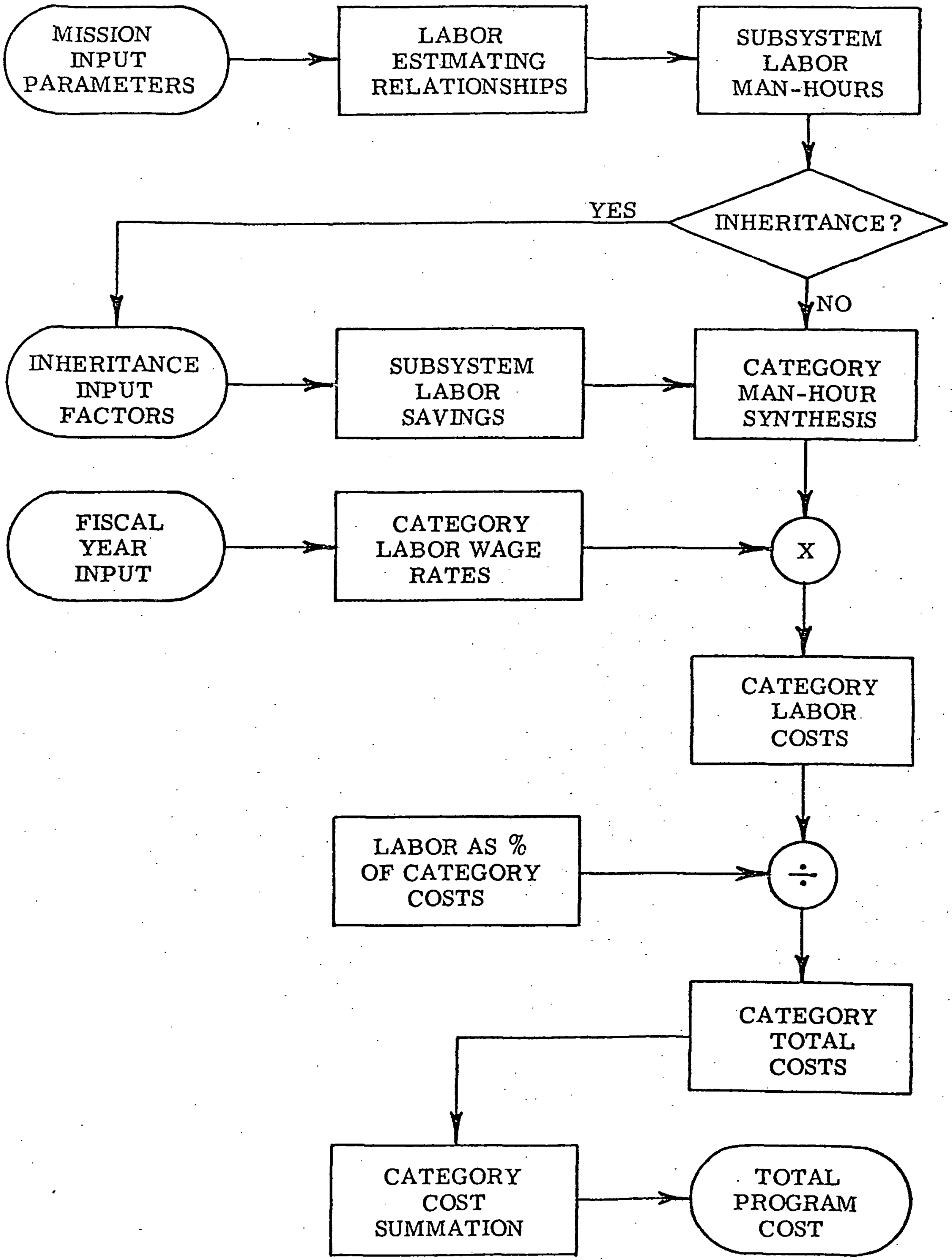

FIGURE 1. PLANETARY COST MODEL SCHEMATIC

The cost model has been more completely summarized in the last Annual Report[1]. The remainder of this summary is devoted to changes and additions to the model which have been accomplished during the past twelve month period. These modifications have been incorporated into a detailed data-sanitized report of the cost model which has just been completed and is included in the bibliography in Section 3.

The cost model analysis during the past year was concentrated on four areas of modelling refinement and on testing through applications. The areas of modification included: 1) updates and additions to the data base; 2) expansion and improvement of the LER's; 3) refinement of the inheritance cost benefit procedures; and 4) refinement of the cost spread analog to reflect the improved characteristics of support category LER's. Each of these improvements are briefly discussed in the following paragraphs.

As part of the continuing cost estimation research at SAI the planetary cost model data base is constantly being updated and expanded. The data base status at the beginning and end of the study period, depicted in Table 2, summarizes these changes. The impact of these updated projects in the data base on labor as a percent of total cost is summarized by hardware and non-hardware categories in Table 3. These data reflect a continued stability in the category labor cost fractions. The largest adjustment in labor as a percent of category cost occurs for the communications subsystem (a hardware category) with a change in the average of only 1.4%.

1. "Annual Report-Advanced Planetary Analysis", Science Applications, Inc. Report No. SAI-120-A1, January 1974.

Table 2

COST MODEL DATA BASE

	DATA BASE STATUS	
	January 1974	January 1975
Surveyor	Complete	Complete
Lunar Orbiter	Complete	Complete
Mariner '64	Complete	Complete
Mariner '69	Complete	Complete
Mariner '71	Updated	Complete
Pioneer 10/11	Complete	Complete
Mariner Venus/Mercury	First Data	Updated
Viking Lander	Updated	Updated
Viking Orbiter	Updated	Updated
Mariner Jupiter/Saturn	First Data	Updated
Pioneer Venus	No Data	No Data

Table 3

LABOR AS PERCENT OF TOTAL COST

	Hardware						
	Structure	Propulsion	G/C	Comm.	Power	Science	Act. Proj. Avg.
Old Data Base Avg.	31.0	26.9	28.6	29.1	29.8	27.7	28.5
Revised/New Prog.							
Mariner Mars '71	31.0	21.2	29.1	31.9	30.0	22.7	27.2
MVM '73	37.8	25.7	32.8	18.8	17.9	21.9	26.2
Viking Orbiter	31.3	20.4	29.6	29.2	31.1	28.9	29.1
Viking Lander	26.7	27.2	29.3	20.6	28.7	24.8	26.2
New Data Base Avg.	31.7	26.5	29.0	27.7	28.7	26.7	28.2

	Non-Hardware								
	Prog. Mgt.	SA.	Test.	OA & R	A. & I.	L/F Ops	Grnd. Eg.	D.A.	Act. Proj. Avg.
Old Data Base Avg.	33.9	33.7	29.8	33.6	32.9	32.6	25.5	26.7	31.1
Revised/New Prog.									
Mariner Mars '71	37.5	31.1	30.7	28.5	40.4	27.6	22.9	23.6	28.1
MVM '73	41.1	35.3	25.3	41.1	42.3	28.0	23.0	-	33.7
Viking Orbiter	38.4	34.1	29.3	30.5	32.9	30.3	22.7	-	31.4
Viking Lander	28.2	42.5	31.3	36.2	24.3	43.2	24.6	-	33.0
New Data Base Avg.	34.7	34.1	29.1	34.2	34.0	31.8	24.8	26.2	31.3

Initial cost model applications indicated the need for some revisions in the Labor Estimating Relationships (LER's) to better model launch and post-launch project costs and to expand the modeling capability to include atmospheric entry probes. Accordingly, the propulsion and power category LER's have been revised for atmospheric probe estimates. These changes, shown in Table 4, are considered preliminary and will probably be further revised as more probe data are accumulated. The new science LER, also presented in Table 4, has slightly smaller coefficients and now only reflects the cost of the instruments. The cost of science teams is now included in a new data analysis LER, presented in Table 5. The original LER for ground equipment and launch/flight operations has been separated into two LER's, one for ground equipment, and the second for launch/flight operations better reflecting the operations cost of longer missions. Both of these new LER's are also presented in Table 5. Factoring the new LER's into the cost model, and reapplying the model to the original eight projects in the data base led to the error summary presented in Table 6. The errors, summarized at the bottom of the table, are slightly larger than earlier results and are due, not to the LER revisions, but the escalating costs of the Viking Lander Program which are now included in the data base. The large estimate errors in both the soft lander programs in the data base, i.e. Surveyor and Viking Lander, are one of the subjects of analysis of the Cost Estimation Research Task during the current contract period.

The characteristics of the inheritance cost benefit procedure of the model are outlined in Table 7. Note that four levels of inheritance are considered. An application of the procedure is presented in Table 8 using the Mariner Venus Mercury Project which relied heavily on inheritance to maximize the spacecraft capability within the design-to-cost

Table 4

REVISED LABOR ESTIMATING RELATIONSHIPS

- PROPULSION LER

$$NR_P^{a} = 21.6(P1)^{1/2}+34.1(P2)^{1/2}+14.4(P3)^{1/2}, \text{ non-probe}$$

$$NR_P = 21.6(P1)^{1/2}+34.1(P2)^{1/2}+11.3(P3)^{1/2}, \text{ probe}$$

$$R_P^{b} = 0.148(N1)(NR_P), \text{ both}$$

where,
- P1 = propulsion system dry weight (lbm)
- P2 = vernier dry weight (lbm)
- P3 = aerodeceleration subsystem weight (lbm)
- N1 = number of flight articles

- POWER LER

$$NR_{EP} = 0.643(W1) + 152, \text{ non-probe}$$

$$NR_{EP} = 0.643(W1) + 50, \text{ probe}$$

$$R_{EP} = 0.15(N1)(NR_{EP})$$

where,
- W1 = non-RTG power weight (lbm)
- N1 = number of flight articles

- SCIENCE INSTRUMENT LER

$$NR_{SE} = 1.5(Q1)+11.5(Q2)+0.105(Q3)+220$$

$$R_{SE} = 0.237(N1)(NR_{SE})$$

where,
- Q1 = total science experiment weight (lbm)
- Q2 = significant lander science weight (lbm)
- Q3 = imaging resolution (pixels/line)
- N1 = number of flight articles

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a. NR_X = Non-Recurring direct labor hours of category X

R_X = Recurring direct labor hours of category X

Table 5

NEW LABOR ESTIMATING RELATIONSHIPS

- GROUND EQUIPMENT LER

$$DLH^{a}_{GE} = 0.033(DLH_{SS}-DLH_{ST})^{1.1} \Big/ \left(1-0.7e^{-D3/2}\right)$$

where, DLH_{SS} = total subsystem direct labor hours

DLH_{ST} = structure direct labor hours

D3 = launch date minus 1971 (zero before 1971)

- LAUNCH/FLIGHT OPERATIONS LER

$$DLH_{LFO} = 90(N1) + 3(K1) + 25(K2)(K3)(K4)$$

where, K1 = mission duration (months)
K2 = total encounter time (months)
K3 = number of encounter start-ups
K4 = total number of science teams
N1 = number of flight articles

- DATA ANALYSIS LER

$$DLH_{DA} = \left[150 + 10(K2)(K3)(K4)\right]\left(1-0.82e^{-D4/3}\right)$$

where, K2, K3, K4 are defined above
D4 = launch date minus 1966.2 (zero before 1966.2)

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a. DLH_X = total direct labor hours of support category X

Table 6

COST MODEL SUMMARY ERROR ANALYSIS

Project	Actual ($M)	Estimated ($M)	% Error
Mariner 64	78.6	74.7	- 5.0
Mariner 69	126.3	110.6	-12.4
Mariner 71	122.4	134.3	9.7
Pioneer F/G[a]	83.8	95.9	14.4
Viking Orbiter	244.3	232.0	- 5.0
Lunar Orbiter	139.2	155.7	11.8
Viking Lander[a]	520.3	392.4	-24.6
Surveyor	420.4	299.2	-28.8

All Projects	Mean Error	=	- 5.0%
	Mean Absolute Error	=	14.0%
Without Surveyor & Viking Lander	Mean Error	=	2.3%
	Mean Absolute Error	=	9.7%

a. RTG's included

Table 7

MODEL INHERITANCE CHARACTERISTICS

- Class One: Off-the-Shelf.

  The subsystem is taken off of the shelf in working condition or ordered while the normal production line is operating as an additional unit.

  - o Inheritance = 100% of non-recurring cost (NRC)
  - o Cost = recurring cost (RC)

- Class Two: Exact Repeat of Subsystem.

  The exact repeat of previous subsystem but to be used in slightly different spacecraft or after line has closed down. Only design work is needed.

  - o Inheritance = 80% of NRC
  - o Cost = 20% of NRC + 100% of RC

- Class Three: Minor Modifications of Subsystem.

  A previous design is required but it requires minor modifications. Thus, the spacecraft will still incur all the design cost and most of the test and development cost in ensuring compatibility of the old design and the new minor mods with the new use of the subsystem.

  - o Inheritance = 25% NRC
  - o Cost = 75% of NRC + 100% of RC

- Class Four: Major Modifications of Subsystem.

  A previous design is required but major modifications have to be made to the design. This gets very close to a new subsystem since even new subsystems rely on previous design and experience. Some savings in development is possible.

  - o Inheritance = 5% NRC
  - o Cost = 95% of NRC + 100% RC

Table 8

INHERITANCE EFFECT ON MVM '73 ESTIMATE

	Class			
• Inheritance Percentages	1	2	3	4
Structure	0	0	50	0
Propulsion	0	50	40	10
Guidance & Control	0	50	25	20
Communications	0	35	50	10
Power	0	25	50	25
Science Instruments	0	20	70	0

• Results

o Actual Cost	96.8 $M
o Estimate Without Inheritance	144.6
o Estimate With Inheritance	93.5

constraint of $100M under which this mission was performed. It can be seen from these data that the assigned inheritance percentages, which are model inputs, need not be particularly accurate to obtain reasonable estimates of total cost and savings.

The final area of analysis performed in this task was refinement in the cost spread analogs which distribute the cost estimate across the years of project performance. The new analogs improve the post-launch run-out cost schedule with the aid of the new data analysis LER. The characteristics of the analogs for both nominal and compressed schedules are summarized in Table 9.

The present cost model has been applied to the planetary mission model covering missions scheduled through the 1980's. It has also already been used several times in advanced planning activities. This initial experience of applications has been encouraging. The cost estimation research is currently being continued to expand the data base and add capability to estimate more ambitious projects such as sample return missions. A task report on the updated cost model entitled "Manpower/Cost Estimation Model for Automated Planetary Projects" has just been completed and, as mentioned above, is included in the bibliography of Section 3.

2.2. Planetary Missions Performance Handbook - Vol. I, Outer Planets (1450 man-hours)

The purpose of the Planetary Missions Performance (PMP) Handbook series is to provide planetary program planners with basic performance data essential in the preliminary steps of mission selection and planning. Two types of NASA handbooks have been generated in the past for planetary mission analysis work: 1) raw trajectory data handbooks such as the NASA SP-35 series, and 2) propulsion system

Table 9

COST SPREAD

- Nominal Spread
  - o 5-year schedule
  - o launch - 3 years to launch + 2 years
  - o launch & flights ops and data analysis excluded

- Compressed Spread
  - o 4-year schedule
  - o launch - 2-1/2 years to launch + 1-1/2 years
  - o launch & flight ops and data analysis excluded

- L/F Operations
  - o launch cost - year launch
  - o cruise cost - amortized
  - o encounter cost - encounter year(s)

- Data Analysis
  - o cost initized 1 month prior to first encounter
  - o spread to 2 years after encounter(s)

performance data handbooks such as the NASA Launch Vehicle Estimating Factors Document. The PMP Handbook represents a marriage of these two basic types of data into a form more directly applicable to mission performance evaluation and planning. The basic format of the PMP Handbook data is net payload versus trip time. Additional data are included to investigate performance sensitivity to such parameters as launch window, swingby distance, orbit size, and navigation impulse budget. Since the basic performance data are sensitive to changes in assumed propulsion capabilities, the Handbook has been organized and assembled in a manner permitting revisions and additions which insure its continued application to mission planning problems.

Volume I of the Handbook deals with payload performance of flyby, swingby and orbiter missions to the outer planets. The scope of missions and launch opportunities covered is defined in Table 10. Note that no data are indicated for S/U/N swingbys in 1984, Jupiter flybys and orbiters in 1982, and Saturn flybys and orbiters in 1984. The launch opportunity spacing for these missions is approximately 13 months so that occasionally a calendar year will not contain a launch opportunity. For Uranus flyby and orbiter missions, data are indicated in only three years: 1980, 1985,and 1990. In this case, launch opportunities do exist in intermediate years but are not presented. The yearly performance changes for Uranus mission opportunities are so small (due to the planet's slow motion around the Sun) that launch year performance dependence can be adequately presented with data from every fifth opportunity.

The propulsion systems used to define payload performance results fall into three classes: 1) launch vehicles, 2) interplanetary

Table 10

DATA SCOPE OF PMP HANDBOOK, VOLUME I

MISSIONS	LAUNCH OPPORTUNITIES (19XX)														
	76	77	78	79	80	81	82	83	84	85	86	87	88	89	90
Jupiter/Uranus/Neptune Swingbys				X	X	X									
Saturn/Uranus/Neptune Swingbys					X	X	X	X		X					
Jupiter Flybys	X	X	X	X	X	X		X	X	X	X	X	X	X	X
Jupiter Orbiters	X	X	X	X	X	X		X	X	X	X	X	X	X	X
Saturn Flybys	X	X	X	X	X	X	X	X		X	X	X	X	X	X
Saturn Orbiters	X	X	X	X	X	X	X	X		X	X	X	X	X	X
Uranus Flybys					X					X					X
Uranus Orbiters					X					X					X

solar-electric low-thrust systems, and 3) orbit retro propulsion stages. The specific periods of application of the various options within each propulsion class for which payload data is presented in the Mission Sections of the Handbook are defined in Table 11. Two base launch vehicles are used in the Handbook. The expendable Titan IIIE during the period 1976-85 and the reusable Space Shuttle during the period 1981-90. There are a number of existing and conceptual chemical upper stages and kick stages which can be used in combination with either of these base vehicles. The upper (and kick) stages chosen for the Handbook data are presented in Table 12. Also given in the table are: the base stage(s) to which each upper stage can be mated, the period of application, and basic propulsion parameters of each stage.

Solar-electric propulsion (SEP) low-thrust systems are the second class of propulsion included in Handbook mission performance data. SEP system selections are representative of stage or propulsion module concepts rather than spacecraft-integrated systems because: 1) a modular concept can be more uniformly applied across the scope of Handbook missions facilitating comparisions with all-ballistic performance data, and 2) standardized systems are much more consistent with the present direction in NASA development programs towards lower cost. Two specific SEP system design concepts are indicated in Table 11: 1) a 20 kw concept for application beginning in 1980, and 2) a "growth" concept rated at 40 kw available in 1985. Key design and performance characteristics of these SEP system options are presented in Table 13. The SEP (20) option parameters reflect present estimates of a current tecynology design. The SEP (40) option parameters presume some degree of technology advance before development. SEP-based payload performance data is presented in the Handbook Mission Sections in the same form of net payload versus trip time as used for

Table 11

PROPULSION SYSTEM APPLICATIONS FOR PMP HANDBOOK, VOLUME I

PROPULSION SYSTEMS CLASSES	LAUNCH OPPORTUNITIES (19XX)														
	76	77	78	79	80	81	82	83	84	85	86	87	88	89	90
1) BASE LAUNCH VEHICLES															
Titan IIIE	X	X	X	X	X	X	X	X	X	X					
Space Shuttle						X	X	X	X	X	X	X	X	X	X
2) SOLAR ELECTRIC SYSTEMS															
20 KW Power Level[a]					X	X	X	X	X	X	X	X	X	X	X
40 KW Power Level										X	X	X	X	X	X
3) RUBBER RETRO STAGES															
Earth-Storable	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
Space Storable					X	X	X	X	X	X	X	X	X	X	X

[a] SEP (20) performance results also presented for the 1979 J/U/N swingby opportunity

Table 12

SELECTED CHEMICAL STAGES FOR PMP HANDBOOK, VOLUME I

Stages	Base Stage(s)	Period of Application	Ignition Mass (kg)	Propellant Capacity (kg)	Stage Mass Fraction	Specific Impulse (sec)
Upper Stages						
Centaur D1-T	Titan IIIE	1976-1985	15620	13540	.867	444
Centaur D1-S	Shuttle	1981-1985	15945	13540	.849	443
IUS-Transtage	Shuttle	1981-1985	16775	14515	.865	311
Tug	Shuttle	1985-1990	25505	22630	.887	456
Kick Stages						
MJS-PM	Centaur D1-T & D1-S	1976-1985	1215	1045	.860	283
3ASK	IUS-Transtage & EE-Kick	1981-1985	2405	1995	.830	288
EE-KICK	IUS-Transtage & Tug	1981-1990	7365	6395	.868	290

Table 13

SEP OPTIONS FOR PMP HANDBOOK, VOLUME I

PROPULSION PARAMETERS	SEP (20)	SEP (40)
Input Power @ 1AU, P_o (kw)	20	40
Power Profile	P/P_o [a]	P/P_o [a]
Thruster Specific Impulse (sec)	3000	3000
Propulsion Efficiency (%)	64	64
Propulsion System Specific Mass, α(kg/kw)	30	20
Propellant Tankage Fraction (%)	3.5	3.5
Support Subsystem Mass (kg)	420	420
Auxiliary Power (watts)	500	1000
Propulsion Time Constraint (days)	350[b]	350[b]
Thrust Direction	-Optimized-	

a. Input power dependence on solar radial distance, R, is given by the following relationships:

$$P/P_o = \begin{cases} 1.4382R^{-2} - 0.2235R^{-3} - 0.2147R^{-4} & \text{if } R \geq 0.68\,\text{AU} \\ 1.3952 & \text{if } R < 0.68\,\text{AU} \end{cases}$$

where P = input power at R AU,

P_o = input power at 1 AU.

b. Reduced to 300 days for J/U/N Swingbys

all-ballistic propulsion systems. Net payload is defined as the injected mass minus the low-thrust propulsion system mass, propellant and tankage, support subsystems, and the chemical retro system (orbiters only). Injected mass degradation due to an extended launch window and high DLA penalties is accounted for in the net payload results.

Retro stages are the third class of propulsion used in the Handbook payload performance computations. Specifically, they are used for orbiter missions, all of which are presumed to require a chemical retro stage for impulsive orbit capture. Orbiter performance data presented in the Handbook are restricted to single stage applications. Multi-stage retro systems are considered unnecessary for the planet approach payload and capture impulse ranges encompassed by the scope of orbiter missions in this volume of the Handbook. Two retro options are considered: 1) a flight-proven bi-propellant earth-storable system with an Isp of 283 sec, and 2) a new, present technology, bi-propellant space-storable system with an Isp of 375 sec. Both options are rubber stages, i.e. the propellant tanks are sized to the specific conditions of planet approach mass and excess speed of each fixed flight time transfer. The relevant parameters for each option are presented in Table 14. The earth-storable stage characteristics are representative of present planetary retro systems such as those used for Mariner Mars 1971 mission and the 1975 Mars Viking mission. The space-storable stage characteristics are indicative of demonstrated technology designs, using FLOX/MMH propellant, which have not yet been developed and flight tested.

Payload performance results and basic transfer characteristics are organized by mission sections in the Handbook. There are eight Mission Sections, one for each mission presented in Table 10. Each section is tabbed and has its own pagination for

Table 14

RETRO STAGE OPTIONS FOR PMP HANDBOOK, VOLUME I

Retro Parameters	Earth Storable	Space Storable
Period of Application	1976-90	1980-90
Retro Engine Mass, M_e	57	66
Tankage Structure Factor, f	0.15	0.16
Propellant Isp (sec)	283	375
Exhaust Velocity, c	2.775	3.677

referencing convenience. Within these sections a consistent pattern of organization is followed. It begins with an introductory subsection which briefly describes the mission, lists the launch opportunities, presents a summary of payload performance sensitivity to launch opportunity, and defines the propulsion options considered for each opportunity. The remainder of the section contains payload performance data organized by launch opportunity.

The specific format and amount of performance data presented varies with the type of mission considered. For flyby missions just one graph is presented for each launch opportunity. It presents the trade-off of net swingby payload versus trip time to the target planet. For swingby missions three graphs are presented for each launch opportunity: 1) net swingby payload curves versus trip time to the second planet, 2) trip time to the first and third planets versus trip time to the second, and 3) swingby miss distances versus trip time to the second planet.

For orbiter missions two types of data are presented for each launch opportunity: 1) graphical performance results of net orbited payload versus trip time for a specific orbit and retro propulsion system, and 2) tabular performance data for each selected combination of propulsion showing the trade-offs between orbit period, trip time, and orbit periapse radius. An example of the tabular data is presented as Table 15. The example chosen is the 1985 Saturn Orbiter mission using the Shuttle/Centaur D1-S/MJS-PM with space-storable retro propulsion. Launch window and excess ΔV allowance are also specified at the top of the table. Two mass results are presented in the body of the table: 1) net useful (orbited) payload in the upper half, and 2) respective retro stage masses in the lower half of the table. Data is presented for three orbit periods (first column). For each period five transfer times are given (second

SATURN ORBITER MASS PERFORMANCE

LAUNCH VEHICLE SHUTTLE/CENTAUR D1-S/MJS-PM
LAUNCH WINDOW 21 DAYS
RETRO SYSTEM SPACE-STORABLE (ISP= 375 SEC)
EXCESS DV 150 M/SEC

*** NET USEFUL PAYLOAD (KG) ***

PERIOD	TRIP TIME		ORBIT PERIAPSE RADII (PLANET RADII)						
(DAYS)	(DAYS)	(YRS)	1.1	2.0	3.0	4.0	6.0	8.0	10.0
15.0	1200	3.29	246	177	127	92	44	12	0
15.0	1400	3.83	389	314	258	216	156	113	80
15.0	1600	4.38	480	408	352	310	245	198	160
15.0	1800	4.93	530	462	409	367	303	254	215
15.0	2000	5.48	546	482	431	391	328	280	241
30.0	1200	3.29	267	201	153	119	72	40	17
30.0	1400	3.83	418	349	297	257	200	159	128
30.0	1600	4.38	514	450	400	361	302	259	224
30.0	1800	4.93	567	508	461	425	368	325	289
30.0	2000	5.48	583	529	485	450	396	355	320
60.0	1200	3.29	281	216	170	136	90	59	36
60.0	1400	3.83	436	372	322	285	230	191	161
60.0	1600	4.38	536	478	431	395	341	300	268
60.0	1800	4.93	591	538	496	463	412	372	340
60.0	2000	5.48	608	560	521	491	443	405	374

*** RETRO STAGE MASS (KG) ***

PERIOD	TRIP TIME		ORBIT PERIAPSE RADII (PLANET RADII)						
(DAYS)	(DAYS)	(YRS)	1.1	2.0	3.0	4.0	6.0	8.0	10.0
15.0	1200	3.29	402	471	521	556	604	636	0
15.0	1400	3.83	390	465	521	563	623	666	699
15.0	1600	4.38	364	436	492	534	599	646	684
15.0	1800	4.93	341	409	462	504	568	617	656
15.0	2000	5.48	323	387	438	478	541	589	628
30.0	1200	3.29	381	447	495	529	576	608	631
30.0	1400	3.83	361	430	482	522	579	620	651
30.0	1600	4.38	330	394	444	483	542	585	620
30.0	1800	4.93	304	363	410	446	503	546	582
30.0	2000	5.48	286	340	384	419	473	514	549
60.0	1200	3.29	367	432	478	512	558	589	612
60.0	1400	3.83	343	407	457	494	549	588	618
60.0	1600	4.38	308	366	413	449	503	544	576
60.0	1800	4.93	280	333	375	408	459	499	531
60.0	2000	5.48	261	309	348	378	426	464	495

and third columns). For each transfer time, seven mass results are presented (columns 4-10), one for each of seven orbit periapse radii. A final section entitled Adjustment Factors is included in the Handbook for computing net payloads for different launch windows, and in the case of orbiters, for different excess ΔV allowances.

Work on Volume I of the PMP Handbook was initiated in the contract period 1 February 1973 to 31 January 1974 and finished during the contract period just completed. The Handbook is contained in a three-ring binder cover in order to facilitate future additions. It is included as one of the distributed contract reports in the bibliography of Section 3. A new task has just been started on Vol. II of the PMP Handbook Series devoted to the inner planets. The first edition of Vol. II will contain performance data for Venus and Mars missions during the period 1980-90, including Mars Surface Sample Return missions.

2.3. Shuttle Impact Planning (500 man-hours)

The purpose of this task was to provide technical assistance and evaluation support to the Planetary Programs Division in monitoring the evolving Space Transportation System (Space Shuttle and upper stages) for its impact on planetary mission planning. Specific areas of concern in interfacing planetary spacecraft with the STS include weight, volume, environment, communications, retro propulsion constraints, launch opportunity dependence, launch windows, and cost benefits. As part of this task SAI was assigned a membership role on the Lunar and Planetary Paylod/Shuttle Working Group (LPP/SWG) which was formed to address issues and problem areas in interfacing automated exploration payloads with the STS.

Three subtasks were performed in this task before it was prematurely terminated to await results of Interim Upper Stage

Contractor Definition Studies which were initiated by NASA in conjunction with the DOD. These subtasks were: 1) development of planetary injection performance requirements (injected payload and C3) for planetary missions planned in the period 1981-85, 2) evaluation of candidate Interim Upper Stages (IUS) for performance capability and cost, and 3) analysis of extended Shuttle Orbiter performance to improve the escape payload capability of smaller IUS candidates.

For the first subtask, eleven missions were analyzed, three of which included alternative solar electric as well as ballistic interplanetary flight profiles. These missions, along with the injected payload performance requirements are presented in Table 16. In comparing these requirements with typical launch vehicle escape performance curves it was found that four missions, the 1985 Venus Buoyant Station, the 1981 Mariner Jupiter Orbiter, the 1985 Mariner Saturn Orbiter, and the 1981 Pioneer Saturn/Uranus Probe were the "driver" missions for IUS selection in this interim period.

The second subtask involved comparing preliminary performance estimates of several proposed IUS candidates with these performance requirements and evaluating their capability and cost in launching these missions. Several important conclusions resulted from this analysis; 1) IUS reusability was a significant cost reduction feature of candidate stage characteristics, 2) smaller IUS candidates, while costing less, led to more substitutions of the more expensive expendable Titan IIIE/Centaur/TE364-4 vehicle on "driver" missions, and 3) a larger IUS candidate could lead to lower cost, heavier payload designs and also relieve NASA of the obligation of an early introduction date for the reusable Space Tug if funding problems occur.

Table 16

TRANSITION PERIOD PLANETARY MISSIONS

Launch Year	Mission	ID	Launches	Flight Time	C3 (km^2/sec^2)	Injected Mass (kg)
1981	Inner Planet Follow-on (Mars)	IPFM	2	300^d	10	1100
1981	Pioneer Saturn/Uranus Probe	PSU	1	3.3^y	142	500
1981	Mariner Jupiter Orbiter (BALL)	MJOB	2	850^d	80	1498
	Mariner Jupiter Orbiter (SEPS)*	MJOS	(2)	800^d	34	2723
1981	Encke Rendezvous (SEPS)	ERS	2	1050^d	43	2198
	Encke Rendezvous (BALL)*	ERB	(2)	1085^d	56	6778
1983	Inner Planet Follow-on (Venus)	IPFV	1	120^d	24	725
1983	Venus Orbit Imaging Radar	VOIR	2	120^d	12	4038
1984	Mars Surface Sample Return	MSSR	2	1018^d	16	3283
1984	Pioneer Jupiter Probe	PJP	2	830^d	88	520
1985	Venus Buoyant Station	VBS	2	175^d	15	5443
1985	Mariner Saturn Orbiter (BALL)	MSOB	2	5.5^y	115	1275
	Mariner Saturn Orbiter (SEPS)*	MSOS	(2)	5.5^y	53	2796
1985	Halley Flyby	HF	1	60^d	17	580

*Alternative Flight Mode

The third subtask was an exploratory analysis, conducted with Rockwell International, to examine the possibility of using the Shuttle Orbiter to place a smaller IUS into a higher energy parking orbit before beginning the escape maneuver. Results from this effort indicated that larger upper stages launched from conventional Shuttle parking orbits, i.e. 160 nm altitude circular, are preferred for planetary missions. As an example, the IUS Transtage was combined with the MJS-PM kick stage to inject the Mariner Jupiter Orbiter payload to escape. Using the conventional Shuttle delivery mode to parking orbit this propulsion combination achieves a C3 of only 62 km^2/sec^2, far less than the 80 km^2/sec^2 required for the ballistic mission (see Table 16). Disregarding a number of important Shuttle Orbiter operational constraints, e.g. reentry heating, the best the Orbiter can do is to raise the C3 capability to 74 km^2/sec^2, still not sufficient to perform the Jupiter Orbiter mission. This Shuttle technique, termed super-orbit injection, was concluded to be of little assistance to high energy escape missions.

As mentioned above, this task was prematurely concluded to await improved IUS performance results. Consequently, no report was prepared for the task. Instead, another task (Task 7 described below) was undertaken to analyze the difficulty of deploying surface penetrators at solar system bodies without atmospheres. This study was particularly relevant to Code SL's advanced planning needs since the penetrator concept had only recently emerged as a potentially useful planetary surface exploration tool. The results of this substitute task are discussed in Section 2.7 below.

2.4. Jupiter Orbiter Lifetime Analysis (650 man-hours)

The four Galilean satellites of Jupiter present a long-term collision hazard to an uncontrolled orbiting spacecraft that repeatedly enters the spatial region occupied by the satellites. An assessment of

this risk and its implication for Jupiter mission planning becomes important if quarantine constraints, currently under review, are imposed on an unsterilized spacecraft. The purpose of this task was to evaluate the likelihood of collision with the Galilean satellites over a wide range of initial orbit conditions with the effect of orbit inclination being of key interest. The scope of the analysis was restricted to orbital dynamic considerations alone, i.e. the question of biological contamination given the event of collision was not addressed. A quarantine or orbiter lifetime of 50 years was assumed. This time period began at spacecraft "shutdown" following completion of the science mission objectives.

A numerical approach was adopted wherein each initial orbit was propagated for 50 years, and satellite closest encounter distances were recorded on every revolution. The computer program developed for this purpose strikes a necessary compromise between orbit computation accuracy and speed. It includes approximations of the three major perturbation effects on the long-term motion of the orbiter: (1) Jupiter oblateness, (2) solar gravity, and (3) satelllte gravity. Program execution time is about 1 minute to complete 600 orbit revolutions typical of a 50-year lifetime. The loss of definitive accuracy in favor of rapid simulation was compensated for by adopting a broad statistical viewpoint regarding the question of collision probability or likelihood. This required the generation of a fairly large number of data samples, a method we refer to as "orbit flooding". It should be noted, however, that this was not a Monte Carlo simulation, which even with the approximate numerical approach used would require a prohibitive amount of computer time.

Numerical data has been generated for 32 basic orbits comprised of 2 perijove distances (5 and 11 Jupiter radii), 2 orbit

periods (21.3 and 60 days), and 8 inclinations between 0° and 90°. The initial epoch for each orbit was sampled over a 7-day interval defined by the characteristic phase resonance (syzygy) of the three inner Galilean satellites, Io, Europa and Ganymede. A sample size of 15 epochs, spaced uniformly 0.5 day apart, was used. All time samples were tacitly assumed to be equally likely. In total then, the Jupiter orbiter space was filled with 480 initial orbits each propagated for 50 years. Significant perturbation of the orbital elements during this time resulted in further permeation of the sampling space.

An overall summary of results is given by the collision record for all satellites presented in Table 17. Of the 480 orbits, the total number of first collision occurences is 81 or 17%. This is of course biased by the equatorial orbit cases; if these are excluded then the first collisions number 34 of 420 orbits, or 8%. The equatorial orbits, representing a worst case upper bound, are physically unreasonable in that the Galilean satellites are not exactly in Jupiter's equatorial plane nor would a spacecraft be placed exactly in this plane. The uniqueness of $I = 0$ is seen by the total number of collisions when orbit continuation is allowed. For example, taking the $5R_J$, 21.3^d orbit, there are an average of 5 subsequent satellite impacts following the first collision. This does not happen when $I \neq 0$. Raising the orbit inclination reduces the risk of collision, yet collisions were recorded even at 60° and 90° inclination. The orbit class having a perijove of $5R_J$ and period of 21.3 days is most susceptible to collision because all satellite orbits are crossed with greater frequency. Io is the dominant body in this case accounting for 50% of the collision occurrences over all eight inclination samples.

Table 17

JUPITER ORBITER COLLISION RECORD

Nominal Orbit Lifetime = 50 years
Initial Epoch Samples/Case = 15
( ) = Total Collisions with Continuation

Number of First Collisions

Orbit Case	$i = 0^o$	$i = 0.5^o$	$i = 1^o$	$i = 5^o$	$i = 10^o$	$i = 30^o$	$i = 60^o$	$i = 90^o$
Perijove = 5 R_J Period = 21.33^d	14 (84)	5	2	4 (5)	2	1	1	1
5 R_J 60^d	14 (51)	4	2	0	0	0	0	0
11 R_J 21.33^d	13 (65)	2	2	2	0	0	0	0
11 R_J 60^d	6 (21)	1	1	2	0	1	1	0

Summary	Number of Initial Orbits	Number of First Collisions
Including Equatorial Orbits	480	81
Excluding Equatorial Orbits	420	34

Fig. 2 summarizes the likelihood of close encounters and collisions taken as an average over all four orbit classes. Graphed as a function of inclination on linear scale, it clearly indicates the rapid decrease between 0^{o} and 10^{o} followed by a leveling off trend. The analytical prediction curve is based on Wetherill's asteroid collision theory applied to the present problem without modification. The comparison serves as corroborating evidence of the basic validity of the numerical data. Discussion of the analytical formula and further comparative results is given in the text. Another means of validation is to examine the ratio of close encounters to collisions. If, for example, this ratio is fractionally small then one would have greater confidence that the event of collision is statistically significant. This was found to be the case.

A general conclusion of this study follows from the summary data shown and other more specific results given in the task report[a]: for the types of crossing orbits investigated, the spacecraft should be placed in an orbit of at least 30^{o} inclination to ensure a 50-year lifetime probability approaching 97-99%. However, if planet and satellite quarantine is imposed on a Jupiter orbiter mission, this lifetime probability may not be high enough. It will then be required to design the post-operational initial orbit specifically for collision avoidance. Among the possibilities mentioned are: 1) hyperbolic escape, 2) circular orbit, 3) critical inclination orbit, and 4) Callisto - resonant orbit beyond Ganymede. The question as to whether such collision avoidance orbits are compatible with the operational sequence and maneuver budget of the nominal mission design was beyond the scope of this study and left for more detailed mission analysis.

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a. "Jupiter Orbiter Lifetime - The Hazard of Galilean Satellite Collision", see bibliography in Section 3.

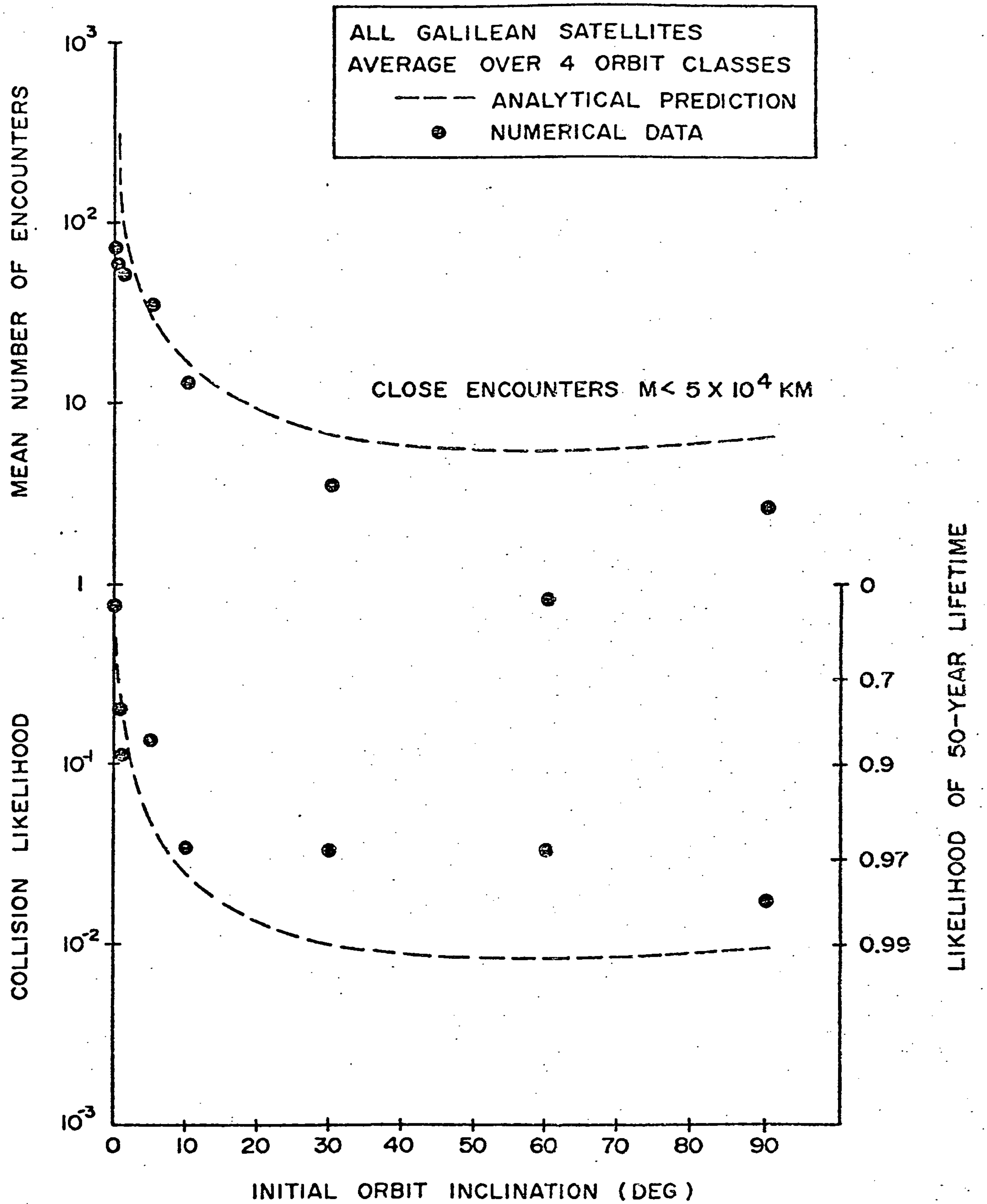


FIG. 2 LIKELIHOOD OF CLOSE ENCOUNTERS AND COLLISION WITH THE GALILEAN SATELLITES FOR A 50-YEAR JUPITER ORBITER LIFETIME

2.5. Titan Mission Concepts Study (490 man-hours)

Titan is the only satellite in our solar system presently known to have an appreciable atmosphere. As such it has long been a body of considerable interest to planetary scientists. In addition to being the smallest known body with an atmosphere, beyond Mars it may be the only atmosphered body whose surface can be reached with an entry probe, and its atmospheric properties have led some investigators to suggest Titan as a possible source of life. Recently, a workshop was sponsored by NASA to assemble and evaluate all available information on the satellite's atmosphere for the purpose of planning future Titan missions. Following the Titan Atmosphere Workshop, this task was assigned to SAI with the objective of generating preliminary definitions of exploratory mission concepts which would serve advanced mission planning needs and provide a basis for selection of more detailed Titan mission studies. Initiated late in the last contract period, this task is being continued as part of the current task schedule with a report of results expected in January 1976.

Four mission concepts are under study: 1) Saturn flybys with Titan atmospheric probes, 2) Saturn orbiters with Titan penetrators, 3) Saturn orbiters with Titan landers, and 4) Titan orbiters. Subjects of consideration for each of these concepts include launch vehicles, launch opportunities, transfer trajectories, spacecraft classes, guidance/ navigation requirements, encounter operations, and data requirements. Analysis to date has been devoted to the earth-Saturn transfer characteristics, Saturn orbit trade-offs, and initial Titan entry studies. Results of this work are beginning to clarify basic Titan mission requirements.

Transfer characteristics have been examined for both flyby and orbiter class missions considering both ballistic and solar-electric low thrust flight modes. For fast ballistic flyby missions

(less than 3.5-year trip times to Saturn) and all solar-electric missions the transfer energy requirements are relatively insensitive to launch opportunity changes. The launch opportunities occur annually being spaced on an average 54-week interval. Ballistic orbiter missions, however, are sensitive to launch opportunity changes with energy requirements modulated by Saturn's movement in and out of the ecliptic plane. The best (minimum energy) opportunity for a Saturn (or Titan) orbiter mission occurs in 1985; the worst opportunity through the end of the century occurs in 1993. Unfortunately, the present pace of outer planet exploration suggests that dedicated Titan missions using Saturn orbits will probably occur closer to 1993 than 1985. Should this prove to be the case, a high energy Shuttle upper stage, e.g. an expendible Tug, and solar-electric propulsion may be required to do these missions.

Minimum Titan asymptotic approach speeds are desired for entry probes, communication characteristics and remote sensing spacecraft experiments. For Saturn-orbiter class Titan missions this creates a dilemma. Net useful payload is increased by lowering the orbit periapse, whereas the Titan approach speed is lowered by raising periapse. It should also be noted that, because Titan is the only massive satellite in the Saturnian system, once a Saturn orbit is established significant changes in the Titan approach speed can only be accomplished with spacecraft propulsion. The orbital Titan approach speed characteristics plotted as a function of trip time to Saturn for the 1985 and 1993 direct ballistic transfers are presented in Fig. 3 for two extreme periapse radii: a low value of 3 Saturn radii just above the Rings, and a high value of 19.5 Saturn radii, just below Titan's orbit. Apparent from the figure is the fact that placing the orbit periapsis up near Titan's orbit radius can reduce Titan approach speeds by as much as a factor of four. The difference in approach speeds due to launch opportunity,

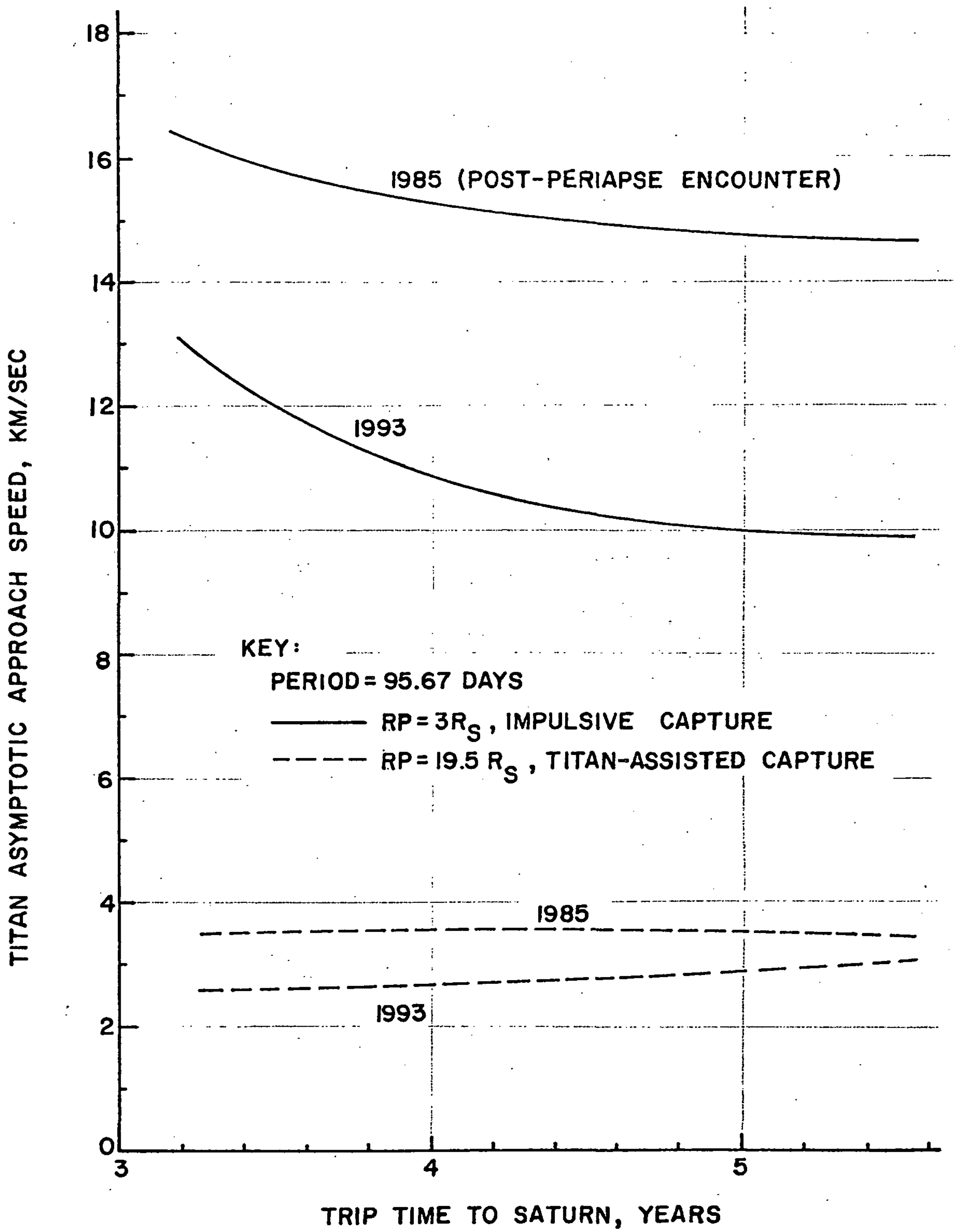


FIG. 3 TITAN ENCOUNTER SPEED COMPARISONS OF SATURN ORBITERS

also shown in the figure, results from the orientation of Saturn's equatorial plane which has an obliquity of about 26.6°.

The performance requirements for establishing a high periapse radius Saturn orbit are rather severe. For example, assume that a 750 kg orbit payload is needed to perform the mission. Using a 2000-day 1993 ballistic transfer, the required injected mass is only 1025 kg if a 3 R_S periapse, 96-day Saturn orbit is selected assuming a space-storable retro propulsion system is used. If instead, a 19.5 R_S periapse radius is selected to achieve the lower Titan approach speeds mentioned above, the injected mass requirement is doubled to 2065 kg. Fortunately, using either a Titan gravity-assist or bielliptic capture maneuver will reduce this requirement considerably and still provide the same orbit. Either of these options will bring the injected mass down to about 1480 kg, which is still a 45% increase over 1025 kg and more than doubles the retro system propellant requirement. It should be noted that the initial orbit period does not influence Titan approach speeds significantly until the period is reduced to less than two Titan orbit periods, i.e. ~32 days. A larger initial orbit period does, however, improve payload performance, which is why a 96-day period (~6 Titan periods) was chosen in the example above. This orbit can be reduced to an elliptical Titan synchronous period with just two Titan swingbys if the orbit inclination and periapse are allowed to float. The Titan approach speed, however, will remain unchanged.

The characteristics of Titan entry trajectories are also being investigated. Initial computations have been done using existing entry probe designs to assess the entry characteristics by comparison with previous study experience. Some preliminary results are presented in Figure 4. Two entry profiles of altitude versus time are presented

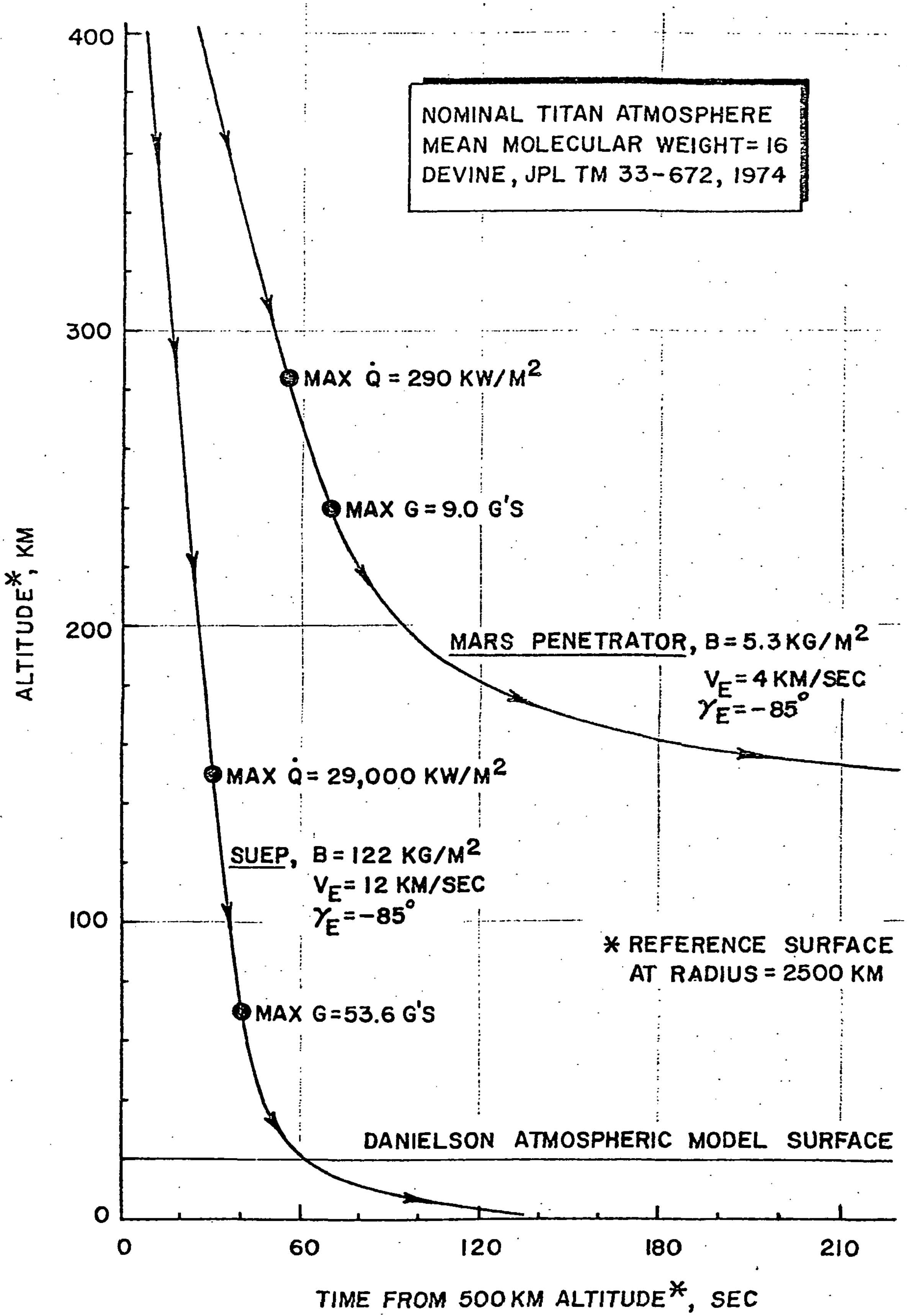


FIG. 4 TITAN ENTRY PROFILES WITH EXISTING HARDWARE DESIGNS

assuming the nominal Titan atmospheric model suggested by Devine[2]. The two profiles can be considered extremes in entry design. The left-hand profile assumes the entry probe is of the SUEP design with a large ballistic coefficient and high entry speed deployed from a Saturn flyby spacecraft. It can be seen that the probe would impact a Danielson-type surface before any data could be collected. On the other hand, if a Pollack-like greenhouse exists this probe might work successfully. The right profile depicts the entry of a Mars-type penetrator with a very low ballistic coefficient and low entry speed typical of a Saturn orbiter deployment. This penetrator entry system is overdesigned for the assumed conditions and tends to "hang up" in the atmosphere well above all postulated Titan "surfaces". Much more work is still to be done on Titan entry, but it is already apparent that the problem is very model dependent and our knowledge of Titan's atmosphere will be a limiting factor on the evolved entry designs.

2.6. Advanced Planning Activity (2000 man-hours)

The purpose of this task is to provide technical assistance to the Planetary Program Office on unscheduled planning activities which arise during the contract period. This type of advanced planning support is a traditional segment of the broader advanced studies work the staff at SAI have performed for Code SL during all past contract periods. The subtasks within this activity range from straightforward exchanges of technical data by phone, through several page responses by mail or telecopier, to more extensive memoranda and presentations, and finally to short mission studies. The level of effort per subtask can vary from as little one man-hour to as much as three man-months. A total of 17

2. Devine, N., "Titan Atmosphere Models (1973)", JPL Tech. Memo 33-672, Jet Propulsion Laboratory, February 1974.

of the more significant advanced planning subtasks, performed during the last contract period, are summarized here, all of which were the subject of written submissions at the time of their completion. These subtasks are tabulated in Table 18. A brief summary description of each of these subtasks is presented in the subsections which follow.

2.6.1. Planetary Mission Cost Estimates With Inheritance

The purpose of this subtask was to provide an initial demonstration of the SAI Cost Estimation Model applied to future planetary missions. Seven missions were analyzed. The missions and their total cost in FY '74 dollars (excluding contingency, NASA management, and contractor fees), including inheritance when applicable, are as follows:

1978	Encke Slow Flyby (SEP)	$139M
1979	Pioneer Saturn Probe	142M
1981	Encke Rendezvous (SEP)	174M
1981	Mariner Jupiter Orbiter	210M
1983	Venus Radar Mapper	214M
1985	Mariner Saturn Orbiter Probe	243M
1987	Mariner Mercury Orbiter	158M

These results do not include launch vehicle or SEP costs. The data were also used to validate other cost estimates of these missions provided as part of a budget planning activity of the then current planetary mission model.

2.6.2. SEP Out-of-Ecliptic Performance Analysis

The purpose of this subtask was to provide a preliminary performance estimate of an SEP IAU out-of-ecliptic mission using either an Atlas/Centaur or Titan IIIE/Centaur launch vehicle.

Table 18

SUMMARY OF 1974-75 ADVANCED PLANNING ACTIVITY

Subtask	Dates	Subject Title	Submitted to
1	Feb.-Mar. 1974	Planetary Mission Cost Estimates with Inheritance	NASA HQ
2	Feb.-Mar. 1974	SEP Out-Of-Ecliptic Performance Analysis	NASA HQ/JPL
3	Apr. 1974	NEP Outer Planet Missions Performance Comparison	NASA HQ
4	Apr. 1974	Summary of Mercury Orbiter Mission Alternatives	NASA HQ
5	Apr.-Jun. 1974	Mars Atmospheric Systems for Exploration Mobility	NASA HQ/JPL
6	May 1974	Ballistic/SEP Outer Planet Missions Performance Comparison	NASA HQ
7	May 1974	Outer Planet Probe Cost Estimates - First Impressions	ARC Probe Workshop
8	May-Jun. 1974	Post-Viking/75 Mars Mission Strategy Analysis	ARC
9	Jun. 1974	1981 Pioneer Mars Penetrator Performance Requirements	ARC
10	Jul.-Aug. 1974	Dual Martian Explorer Mission Concept Evaluation	NASA HQ/LaRC
11	Aug. 1974	Planetary Mission Opportunities Performance Summary	COMPLEX

Table 18 (Cont'd.)

SUMMARY OF 1974-75 ADVANCED PLANNING ACTIVITY

Subtask		Subject Title	Submitted to
12	Aug. 1974	1978/1980 Pioneer Venus Opportunities Comparison	NASA HQ
13	Sep.-Oct. 1974	Planetary Mission Model Cost Estimates	NASA HQ
14	Nov. 1974	Uranus Flyby Launch Vehicle Requirements: 1979-94	SOPE
15	Dec. 1974	Cost Model Description Memo	MSFC/JPL
16	Jan. 1975	OUTLOOK Planetary Mission Cost Estimates	NASA HQ/ OUTLOOK
17	Jan. 1975	PJO_p Type II Transfer Characteristics	SOPE

Performance data were generated in terms of heliographic latitude and net payload mass as a function of launch vehicle, SEP installed power and number of thrust periods (mission time). The results show that a 250 kg net payload can be delivered in 800 days (5 thrust periods) to: 1) a heliographic latitude of about 27° with an Atlas/Centaur/SEP (6 kw); or 2) a latitude of about 54° (twice as inclined) with a Titan IIIE/Centaur/SEP (15 kw). The study results were given to NASA Hq. and JPL, where further analysis confirmed these results and determined some performance improvement with longer thrust periods.

2.6.3. NEP Outer Planet Mission Performance Analysis

The purpose of this subtask was to summarize the performance capability of a 120 kw Nuclear Electric Propulsion (NEP) system compared to several chemical and SEP propulsion options for difficult outer planet missions. Three missions were considered: 1) a 1990 Ganymede Orbiter/lander, 2) a Uranus Orbiter, and 3) a Neptune Orbiter. The results showed that, particularly for the latter two missions, that fewer stages and less trip time are required to deliver equal payloads to orbit if an NEP system is used. These data were supplied to NASA Hq. for the purpose of a presentation on the benefits of nuclear low thrust propulsion for advanced missions in the last decade of this century.

2.6.4. Summary of Mercury Orbiter Mission Alternatives

For advanced program planning purposes a performance comparison of alternative Mercury Orbiter missions was requested. Four Mercury orbiter transfers were analyzed: 1) a 1987 direct ballistic transfer, 2) a 1988 single Venus swingby ballistic transfer, 3) a 1978 double Venus swingby transfer,and 4) a direct SEP (20 kw) transfer. For each type of transfer payload performance data were

computed assuming either a Titan IIIE/Centaur or Shuttle/Centaur launch vehicle and either a circular or elliptical 24-hour Mercury orbiter. Net orbit payload ranged from zero for a direct ballistic transfer to almost 1000 kg for a SEP (20 kw) transfer off the Shuttle/ Centaur. These results were tabulated and transmitted via telecopier in response to an immediate request for the data.

2.6.5. Mars Atmospheric Systems for Exploration Mobility

This subtask was undertaken as a result of a request from the Administrator's office regarding the feasibility of atmospheric devices for future Mars exploration. Working with the Planetary Programs Office a set of five concepts applied to two exploration purposes were defined for analysis. The concepts included aircraft, helicopters, balloons, dirigibles, and surface sailers. The exploration purposes were for atmospheric (altitudes) studies and for transport (range) operations. Of the concepts analyzed, aircraft appeared to have the best application poential. Active lift was considered more useful than buoyancy in the thin Mars atmosphere. All of the concepts had large size/mass ratios, also due to the thin atmosphere. Nothing in earth atmospheric devices would be directly applicable to similar Mars objectives.

2.6.6. Ballistic/SEP Outer Planet Missions Performance Comparison

The purpose of this task was to develop performance comparisons of several "driver" outer planet missions using candidate IUS escape stages with and without solar electric propulsion. The request by NASA Hq. for this data was made to determine if smaller IUS candidates augmented with an SEP stage would provide adequate performance for the more difficult outer planet missions. Net payload

versus flight time performance data were generated for 1) a 1981 Jupiter orbiter, 2) a 1985 Saturn orbiter, and 3) a 1986 Uranus/Neptune swingby. For comparison the transtage and Centaur IUS candidates were used with and without a 20 kw SEP stage. The SEP stage does improve the payload performance of the orbiter missions to acceptable levels at somewhat longer trip times, but cannot meet the high energy requirement of the Uranus/Neptune swingby mission. The results of this analysis were telecopied to NASA Hq. in the form of payload/ flight time performance plots.

2.6.7. Outer Planet Probe Cost Estimates - First Impressions

This subtask was an invited paper requested by ARC with NASA Hq. concurrence for the Outer Planet Probe Technology Workshop held at ARC in May 1974. The purpose of the paper was to examine early estimates of outer planet atmospheric probe cost and evaluate them by comparison with past cost experience of planetary projects. The SAI cost model was heavily involved in this analysis. Using newly derived estimating relationships for planetary entry probes a cost estimate of $48M (FY'74 dollars) was derived compared to a contractor Phase B estimate of $40M. In both estimates the subsystem cost drivers were for science and communications. Savings in attitude control (which is passive) were found to be offset by difficult packaging of components in the probe. The cost of the aero deceleration system was a reasonable fraction of the total cost, but might not be if entry conditions are allowed to exceed the simulation capacities of current and near-future test facilities. The most important point stressed in the paper was the need for more project cost data to improve confidence in cost estimates of future probe missions.

2.6.8. Post-Viking/75 Mars Mission Strategy Analysis

The purpose of this subtask was to address the question: "What type of mission would be a logical follow-on to the Viking/75 lander presuming several different Viking Achievement Scenarios?" The motivation for this study was to determine under what circumstances a Pioneer Mars Penetrator mission might be most preferred for the 1979 Mars launch opportunity. The analysis was requested by ARC as part of their penetrator study activity and had NASA Hq. concurrence. The type of achievement scenarios envisioned ranged from the lunar-type results of Surveyor to detection of active surface life. It was concluded that penetrators would be most popular if Viking results failed to detect life but did find evidence for active internal processes. Penetrators could then address the nature and source of this internal activity as well as extend the search for life related conditions such as subsurface water. It was further concluded that if the Viking results make a strong case for life, the penetrator concept might only be postponed, rather than dropped, until Mars geology reestablished its relevance as an exploration objective.

2.6.9. 1981 Pioneer Mars Penetrator Performance Requirements

The purpose of this task, requested by ARC, was to determine the energy requirements of a 1981 Pioneer Mars Penetrator Mission as part of the contingency planning of the 1979 mission. Type II transfer conditions were found with similar C3 launch requirements, but the Mars approach speeds are up about 400 m/sec. from the 1979 minimum Vhp's of about 2.65 km/sec. The impact of this increase is the requirement for a larger non-existing solid retro motor or removal of one penetrator to decrease the captured orbital mass. A summary of the energy requirements was plotted as a function of launch date and transmitted to ARC for planning considerations.

2.6.10. Dual Martian Explorer (DME) Mission Concept Evaluation

The purpose of this subtask was to evaluate the DME mission concept proposed by the Aeronomy Section of the Planetary Physics Branch at LaRC. SAI was requested by NASA Hq. to make this evaluation since it had recently completed a conceptual study of a Pioneer Mars aeronomy mission and could make an objective comparison of the two concepts. The DME mission involves mother and daughter spacecraft in coplanar Mars orbits simultaneously performing complementary measurements of the thermosphere and exosphere. Measurement objectives include neutral composition, vertical structure, lateral variations, exospheric temperature, and atmospheric energy response. The concept would be based on the Dual Air Density (DAD) Explorer mission to be flown at earth in 1975. A new mother spacecraft would be needed but the daughter would be a modified version of the DAD daughter. A total launched mass of 309 kg is within the capability of the Delta 2914 launch vehicle for either the 1977 or 1979 Mars opportunities. The LaRC estimated cost for this mission excluding DSN services and launch vehicle costs was just under $25M in FY 73 dollars.

The analyses performed verified the basic mission characteristics and developed an independent estimate of cost. In general, the DME mission was considered an interesting alternative to the Pioneer Mars Aeronomy mission. Specific advantages included: 1) no sterilization requirement; 2) pole to pole latitude coverage; 3) dual altitude measurements; and 4) reduced thermal and attitude control loads on the spacecraft. Disadvantages found in the comparison were: 1) greater operational complexity; 2) new spacecraft development for the mother orbiter; and 3) no in situ low altitude (<120 km) science. The mission cost was estimated with the SAI Cost Model to be about

$55M which strongly suggested that the La RC estimate of $25M was much too low. Finally, the science rationale of a Mars aeronomy mission was not yet subjected to competitive planning with other future Mars mission concepts. This was recommended in order to determine at what cost the aeronomy mission ceases to be of competitive value.

2.6.11. Planetary Mission Opportunities Performance Summary

This subtask, performed at the request of NASA Hq. , was a compilation of planetary mission energy requirements and launch vehicle performance curves intended as supporting data for the SSB COMPLEX meeting at Snowmass in August 1974. Injected payload performance curves were included in the data package for the Titan IIIE/ Centaur D1-T and the Shuttle IUS candidates Transtage, Agena, and Centaur along with various added kick stages. Basic transfer characteristics for Venus orbiter and Mars surface sample return missions were provided for launch opportunities between 1979 and 1986. Launch energy requirements were also plotted for outer planet missions to Jupiter, Saturn, Uranus and Neptune during the period 1978-1985 using fixed trip time transfer conditions. These data were used by COMPLEX to evaluate the impact of moving mission launch dates to alternative opportunities as various mission model strategies were explored.

2.6.12. 1978/1980 Pioneer Venus Opportunities Comparison

This short subtask was performed for NASA Hq. as part of their contingency planning for future projects. The objective of the analysis was to evaluate the performance impact of deferred Pioneer Venus launches. Ballistic transfer characteristics of the 1980 opportunity are such that Type II transfers are preferred for both the probe and orbiter missions with both launches taking place in a 21-day period in April 1980. An added kick stage may be required for the probe

mission and a larger retro motor would be needed for the orbiter due to higher approach speeds. In general, the 1980 transfers would have an unfavorable impact on the Pioneer Venus propulsion requirements, all other systems being equal.

2.6.13. Planetary Mission Model Cost Estimates

Seven missions from the 1973 Planetary Mission Model were evaluated with the SAI cost model as a subtask for NASA Hq. to provide independent cost estimates for evaluation purposes. The missions analyzed were as follows:

Venus Orbiter Imaging Radar	$220M
Mariner Jupiter Orbiter	256M
Encke Rendezvous (SEP)	201M
Pioneer Saturn Probe	108M
Pioneer Saturn/Uranus Flyby	133M
Pioneer Jupiter Probe	126M
Mars Surface Sample Return	690M

where the mission costs are given in FY '75 dollars. Several of these missions are the same as those estimated in Subtask 1. A comparison of costs will show higher values here due to several factors. These are: 1) different mission definitions; 2) improved estimating relationships of post-launch operations and data analysis costs; and 3) FY '75 dollars instead of FY '74 dollars. In general, these estimates compared favorably with those supplied by the various study teams at the NASA Centers responsible for the mission definitions.

2.6.14. Uranus Flyby Launch Vehicle Requirements: 1979-94

The purpose of this task was to prepare and present a summary of performance characteristics of Uranus flyby mission in the

period 1979-1994. The scope of the analysis included four payload levels, six launch vehicles (some including SEP), and swingby as well as direct transfers. Graphical and summary tabular data of flight time requirements with each payload/launch vehicle combination were prepared. The presentation was made to the Symposium on Outer Planet Exploration (SOPE) which was debating the importance of an early Uranus flyby (with optional probe) mission.

Among the conclusions drawn from this summary analysis, it was pointed out that the 1979 Jupiter swingby mission was a unique opportunity in that the propulsion requirements were much less than direct transfers and Uranus was encountered when its pole was facing the sun. If and when an expendable Tug was added to the Space Transportation System, its performance shortened trip times to Uranus more significantly than any other option compared with a Titan/Centaur/Kick direct launch to the planet. The next low energy Jupiter swingby opportunities after the 1979/80 pair begin in 1994. Finally, it was concluded that Vega-class Uranus transfers were impractical, both in terms of total trip time and post-launch maneuver requirements.

2.6.15. Cost Model Description Memo

Several requests were received late in calendar 1974 from MSFC and JPL for details of the SAI cost model to assist in several mission planning activities. Because a publishable document of the model was not scheduled until the end of the contract period (January 1975) a memo was prepared and mailed to these groups indicating the input requirements, computational methodology, and estimating the capability of the cost model. These preliminary data are being followed up with an expanded distribution of the sanitized cost model report referenced in the bibliography of Section 5.

2.6.16. OUTLOOK Planetary Mission Cost Estimates

The purpose of this task was to provide preliminary cost estimates of advanced lunar and planetary missions being considered as part of the long-range planning activity of the NASA-organized OUTLOOK Committee. A total of 30 different missions were analyzed ranging from fairly simple interplanetary probes to complex sample return concepts. Key project event times and baseline performance requirements were generated in addition to the project cost estimates. The estimates were broken down into six categories: science, spacecraft, mission operations, MCCC, NASA management, and contingency. A funding spread of the total cost was also provided based on the assigned launch date. Launch vehicle selections were made for each mission but their costs were not included in the estimates. The total accumulated cost for all 30 missions which span the remainder of this century was almost $9B in FY '75 dollars. No attempt was made in the cost estimation analysis to reflect anything more than typical inheritance cost benefits. Low cost standardized hardware or block buys, for example, were not included in the estimates of project costs.

2.6.17. PJO_p Type II Transfer Characteristics

At the request of the Chairman of the Symposium on Outer Planet Exploration (SOPE) an analysis was performed and presentation given on the encounter characteristics of Type II transfers for a 1980 Pioneer Jupiter Orbiter with probe (PJO_p) mission. A preliminary PJO_p baseline mission definition used a Type I Jupiter transfer which required a nightside entry of the probe. A dayside entry is much preferred scientifically and could be achieved with a longer trip time Type II transfer. The analysis, supported by data received from ARC, showed that a 1050-day Type II transfer (about 200 days longer than the

previous Type I baseline) would permit the entry and a 30 minute descent to be completed in daylight. Of additional benefit was the fact that the approach speed was reduced by about 850 m/sec reducing the orbit capture impulse requirement for the retro propulsion system.

In addition to the Type II PJO_p results, data were also presented for two other outer planet missions of planning interest. The first is a 5.5 year 1983 ballistic Saturn transfer which passes through the lagging Trojan asteroid group. The second is a 1980 Jupiter/Saturn swingby transfer on which the Titan IIIE/Centaur/TE364.4 has sufficient performance to place two Pioneer spacecraft with probes. One could be targeted for a nightside Jupiter entry and the second could perform a terminator entry at Saturn.

2.7. Error/Control Analysis of Penetrator Deployment at the Moon and Mercury (730 man-hours)

Penetrators are missile shaped objects designed to implant electronic instrumentation in a wide variety of soils with a high speed impact, i.e. 150 m/sec. They have been used successfully in many terrestrial applications over the past decade. Recently they have also been proposed for post-viking/75 Mars exploration. The most significant advantage of planetary penetrators is that they avoid the high cost of soft landers without imposing the extreme impact conditions of rough surface landers on the payload. An initial favorable response by the science community to the exploration potential of Mars penetrators has prompted an interest in the application of this concept to in situ subsurface studies of other terrestrial bodies and planetary satellites. Unlike Mars many of these objects do not have atmospheres. A first order feasibility question has thus arisen: "Can penetrators be successfully guided to required near-zero angle-of-attack impact conditions in the absence of an atmosphere"? A preliminary answer to this question was the purpose of this task.

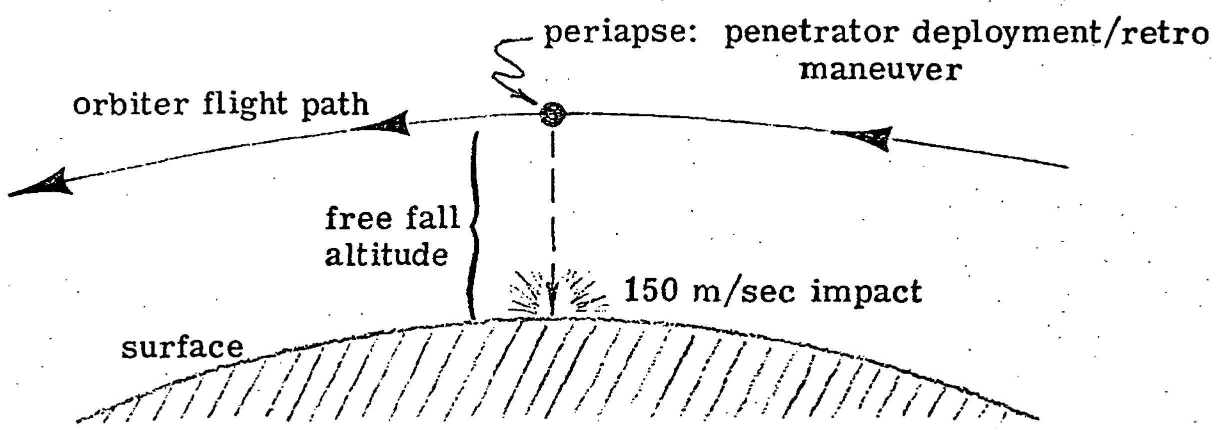
The scope of the analysis included two potential targets, the moon and Mercury, involved several different penetrator deployment modes, and focused on impact errors arising from open-loop and closed-loop deployment control systems. Successful penetrator implacement requires: 1) that the impact speed be controlled, nominally to 150 m/sec; 2) that the penetrator angle of attack, measured between the longitudinal axis and velocity vector, be in the range 0^{o} - 11^{o} at impact; and 3) that the impact flight path angle be within 15^{o} of vertical. It was the errors in these terminal conditions that were the principle concern of this study.

The best mode of penetrator deployment identified uses an orbiting spacecraft as a penetrator launch platform. This mode, labeled the Intermediate Ellipse Transfer (IET) Mode, is depicted in Table 19. Prior to deployment the orbiter is first placed in an elliptical low-periapse altitude orbit. The penetrator is launched at periapse with a retro motor which kills its orbital velocity. It is then pitched over and allowed to free-fall to the surface. The deployment characteristics of the IET Mode are also summarized in Table 19. The initial orbit is circular at the moon and elliptical at Mercury. A free-fall impact velocity of 150 m/sec means that the periapse altitudes of these orbits must be lowered to 7 km and 3 km at the moon and Mercury, respectively, prior to penetrator deployment. This should not be a problem at the moon, but at Mercury a combination of several orbital maneuvers, onboard radar altimetry, and solar perturbation control will be necessary to achieve the very low altitude of 3 km. Also it will be undesirable to leave the orbiter in this orbit for more than several revolutions due to the impact hazard without continuous control. The penetrator retro ΔV requirements to kill the orbital periapse velocity are a nominal 1700 m/sec at the moon, but over 4 km/sec at Mercury. Assuming a

Table 19

REFERENCE DEPLOYMENT MODE

- INTERMEDIATE ELLIPSE TRANSFER (IET) DEPLOYMENT SCHEMATIC



- IET DEPLOYMENT CHARACTERISTICS

	Moon	Mercury
Initial orbit periapse altitude (km)	100	600
Initial orbit eccentricity	0.0	0.8
Penetrator deployment periapse altitude (km)	7.0	3.0
Penetrator retro impulse (m/sec)	1698	4065
Penetrator impact velocity (m/sec)	150	150
Penetrator impact mass (kg)	31	31
Penetrator deployment mass[a] (kg)	81	251

a. Just prior to single stage solid retro.

penetrator impact mass of 31 kg (i. e the Mars design), 10 kg for the attitude control system, and a single stage solid motor retro system, the total deployed mass of each penetrator is 81 kg at the moon and 251 kg at Mercury. These values can be compared with a deployed mass of only 45 kg at Mars where atmospheric braking is used to slow the penetrators.

Key results of the deployment error analysis are summarized in Table 20. The errors in DSN tracking of the orbiter's state at deployment are small and have little effect on any of the impact conditions except impact location. The primary error source of impact velocity and angle-of-attack errors is penetrator retro execution errors. The execution errors shown in the table are scaled to the magnitude of the impulse assuming 3σ pointing errors of 1.5° and 3σ magnitude errors of 1%. The affect of these errors on impact conditions are shown as open-loop impact errors. The critical errors are in impact angles of attack which are dominated by errors in the terminal flight path angle. With a maximum acceptable impact angle of attack of 11° required to successfully penetrate even very soft soils it is readily seen that the open-loop control mode is unsatisfactory having 3σ values of 15 degrees at the moon and 36 degrees at Mercury. Adding an accelerometer triad to the penetrator to monitor the retro burn errors easily reduces the angle of attack errors to very small values (2°) as can be seen by the tabulated closed-loop error summary. It should be noted, however, that nothing is done in the closed-loop mode to correct the execution errors; the attitude control system just accomodates them. Hence, the impact flight path angle, at Mercury in particular, may still be larger than the 15° off-vertical limit desired by some of the penetrator experiments, e. g. seismometers.

Table 20

PENETRATOR DEPLOYMENT ERROR SUMMARY[a]

	Moon	Mercury
• DSN TRACKING ERRORS AT DEPLOYMENT		
Altitude (m)	100	150
Velocity (m/sec)	10	24
• RETRO EXECUTION ERRORS (m/sec)		
Radial ($\dot{x}$)	30	71
In Path ($\dot{y}$)	26	61
Cross Path ($\dot{z}$)	30	71
• OPEN LOOP IMPACT ERRORS		
Speed (m/sec)	8	41
Angle of Attack (deg)	15	36
Miss Distance (km)	15	21
• CLOSED-LOOP IMPACT ERRORS		
Speed (m/sec)	8	41
Angle of Attack (deg)	2	2
Miss Distance (km)	15	21

a. 3σ errors of IET deployment mode

As an overall conclusion to this analysis, the deployment of lunar penetrators appears to pose no unreasonable performance or control requirements. Conversely, the low deployment altitudes, the large retro mass, and large retro execution errors all raise feasibility questions for a Mercury penetrator mission. More detailed analysis will be required to resolve these issues and is recommended.

3. REPORTS AND PUBLICATIONS

Science Applications, Inc. is required, as part of its advanced studies contract with the Planetary Programs Division, to document the results of its analyses. This documentation traditionally has been in one of two forms. First, reports are prepared for each scheduled contract task. Second, publications are prepared by individual staff members on subjects within the contract tasks which are considered of general interest to the aerospace community. A bibliography of the reports and publications completed during the contract period 1 February 1975 through 31 January 1975 is presented below. Unless otherwise indicated, these documents are available to interested readers upon request.

3.1. Task Reports for NASA Contract NASW-2613

1. "Manpower/Cost Estimation Model for Automated Planetary Projects", Lawrence D. Kitchen, Report No. SAI 1-120-194-C1, March 1975.

2. "Planetary Missions Performance Handbook-Volume I, Outer Planets", Report No. SAI 1-120-194-M2, November 1974.

3. "Jupiter Orbiter Lifetime - The Hazard of Galilean Satellite Collision", Alan L. Friedlander, Report No. SAI 1-120-194-T2, February 1975.

4. "Error Analysis of Penetrator Impacts on Bodies Without Atmosphere", Donald R. Davis, Report No. SAI 1-120-194-T3, March 1975.

5. "Advanced Planetary Studies Second Annual Report", Report No. SAI 1-120-194-A2, March 1975.

3.2. Related Publications

1. "Comet Encke Flyby - Asteroid Rendezvous Mission", Alan L. Friedlander, Journal of Spacecraft and Rockets, 11, 4, pp. 270-272, April 1974.

2. "Outer Planet Probe Cost Estimates - First Impressions", John C. Niehoff, Outer Planet Probe Technology Workshop paper, ARC/NASA, May 1974.

3. "Comparison of Advanced Propulsion Capabilities for Future Planetary Missions", John C. Niehoff and Alan L. Friedlander, Journal of Spacecraft and Rockets, 11, 8, pp. 566-573, August, 1974.

4. "Pioneer Mars 1979 Mission Options", John C. Niehoff and Alan L. Friedlander, AIAA Paper No. 74-783, Mechanics and Control of Flight Conference, Anaheim, August 1974.